SOLAR PROBE TECHNOLOGY CHALLENGES

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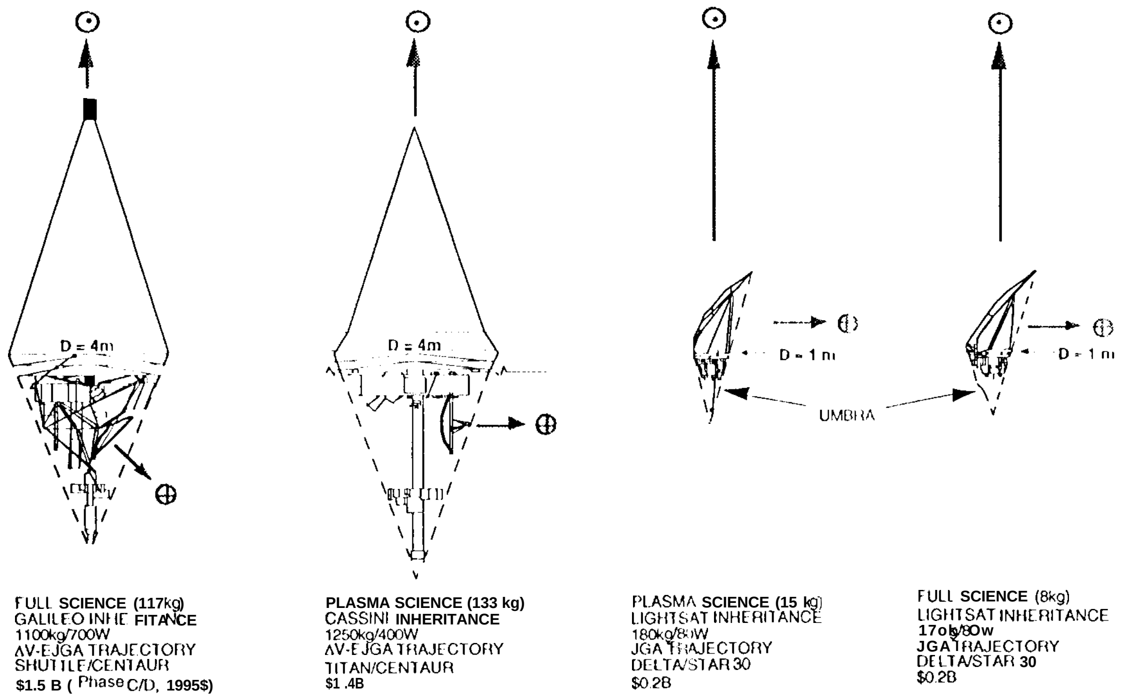
Abstract

A mission close to the sun is only possible if new spacecraft technologies can be developed and incorporated into a state- of-the-art spacecraft concept. The perihelion goal of 4 solar radii requires a shielded spacecraft that can tolerate the 3000 suns solar flux while maintaining the electronics components at room temperature. in addition, the shield surface should sublimate at a rate of less than 3 mg/s at perihelion. Many shield configuration designs have been studied and the most promising is a parabolic shape that functions as both a shield and a large high gain antenna. The shield material chosen for this design is a carbon-carbon material with highly emissive surface properties. A mission requirement for a high telecommunications power sterns from the expected interference when attempting to transmit data through the solar corona. It is expected that the large carbon-carbon shield/antenna will have a large power gain even at high temperatures and will return adequate telemetry at the X-band radio frequency chosen for the Solar Probe mission. Other key technology needs include a non-nuclear power subsystem that can function in the extreme environments of the mission from Earth to Jupiter and onward to a 4 solar radii perihelion.

INTRODUCTION

The Solar Probe mission concept is based on the goal of measuring as far into the solar atmosphere (solar corona) as possible with in-situ instruments. The depth that can be achieved depends primarily on the design of a thermal shield that will protect the sensitive electronics and instruments from the extreme photon flux near the sun. Early studies (Randolph, 1978) determined that with reasonable extrapolations of materials technologies, the spacecraft could reach to a perihelion radius of four solar radii. The science' community (Neugebauer, Davies, 1978) realized that significant new information about the birth and acceleration of the solar wind could be determined from such a close approach to the sun.

A concern expressed early by the science advisors was that the mass loss from outgassing or sublimation from the shield surfaces must be low enough to prevent a self-induced plasma cloud around the spacecraft at a time when instruments were attempting to measure natural plasmas around the sun. Thus, the principal requirement on the shield design was to minimize the mass loss from the shield's surface at perihelion. An initial analysis (Goldstein, et al, 1980) first



a. STARPROBE (1982) b. Solar Probe (1 989) c. Solar Probe (1 994) d. Solar Probe (1995)

Figure 1. Evolution of Solar Probe Spacecraft Concepts (Perihelion Configuration)

identified an acceptable specification for this mass loss of less than 3 mg/s to prevent interference with the scientific measurements. Early shield configurations and materials were conceived in the early 1980s and evolved into a large conical shield concept utilizing carbon-carbon material for the STARPROBE study (Randolph, 1981) that is shown in the left panel of Figure 1. This early shield was a truncated cone that had a narrow cone angle to provide a low perihelion temperature and minimum mass loss. (The cone contained a central optical path for imaging instrument viewing.) It was estimated that the mass loss rate was about 2.5 mg/s at perihelion for this configuration. In addition, this shield was large enough to cast a shadow or umbra over the entire electronics bus and over the high gain antenna within the umbra (shown with the arrow toward the earth). A later version of the spacecraft (shown in panel b of Figure 1) had a large conical shield with no optical path, satisfied the mass loss specification , and satisfied the requirement to contain the antenna within the umbra (Randolph, 1989).

A major change in the shield was conceived in the 1994 configuration shown in panel c of Figure 1. It was recognized that the shield material (Carbon- Carbon) had a high enough electrical conductivity that the material could be used for a parabolic radio antenna. Also, the progress of off-axis parabolic antenna development in industry suggested that an off-axis design might be combined with the thermal shield to produce a multifunctional structure as shown in panel c of Figure 1. The key to this configuration is a quadrature orbital geometry at perihelion (Randolph, 1994) that constrains the earth and the sun to be orthogonal as suggested in the panel by the arrows. This new parabolic shield configuration significantly reduced the spacecraft size as shown in the comparison in Figure 1. Even though the upper end of the parabolic shield would have a higher temperature than the earlier concepts, the total mass loss would be less than 3 mg/s. A new variation in the small parabolic shield was introduced briefly in 1995 (see panel d in Figure 1) to accommodate optical instrument requirements for continuous viewing of the solar disc. Here the lower part of the parabola is replaced with a conical section of shield for the instrument viewing paths. Details of this arrangement are discussed below.

Jj!M1's]I11:1 ,D TECHNOLOGY ISSUES

The thermal shield subsystem must function in diverse thermal environments including launch, 0.04 earth suns at Jupiter, and 2.900 suns at its closest approach to the sun at a perihelion of 4 solar radii. A fundamental] requirement is that the sublimation from the shield surface at perihelion should be less than 3 mg/s (Goldstein, et al, 1980). Previous studies (Randolph, 1978, 1981, 1989, 1994) have identified carbon-carbon as the optimum material for the primary thermal shield. Carbon-carbon has been selected because! of its high strength to weight ratio at temperatures up to 2800 K, low vapor pressure, relatively stable ratio of solar absorptivity to emissivity (α/ε), stability in charged particle and high ultraviolet flux environments, and the depth of the experience with the manufacture and characterization of this general class of materials.

Other materials considered and rejected include tungsten alloys and high temperature ceramics. The tungsten alloys arc heavy and are expected to degrade to the charged-particle flux experienced at both Jupiter and the near-solar corona environment. Of particular concern is '(darkening" of the tungsten resulting in an increase in α/ε resulting in a concomitant increase in the operating temperature of the heat shield at perihelion. In addition the absorption of charged proton particles may result in an increase in the brittle-ductile transition temperature of the tungsten above the operating temperature of the heat shield at Jupiter. This could result in the structural failure of the shield should it be impacted by even a small micrometeriod during the spacecraft passage back to the sun. Although ceramics have the potential to survive at the perihelion temperatures required, there are many technology issues that must ,be addressed before this material class might be considered a prime candidate. The material would have to be reinforced with continuous fibers in order not to experience brittle failure at micrometeriod impact. These materials have much less technology development than carbon-carbon. Only limited thermal and mechanical characterization data is available and a manufacturing capability for the extremely thin, large shell structure required for the thermal shield is not available.

The current carbon-carbon shield design has evolved over the past 20 years to a parabolic surface supported by a number of carbon-carbon thin-wall tubes. This shape supports both the requirement to shield the spacecraft from the direct radiation from the sun during perihelion passage and to provide a high-gain antenna for real time transmission of the scientific data gathered at perihelion. The current heat shield overall size is 2.8 m high by 1.5 m wide. Nominal shield thickness is 0.8 to 1.5 mm. Final shield thickness will be determined by sizing for the launch acceleration and acoustic loads, the maximum "antenna" distortion acceptable for telecommunications, and the requirement to reduce the extent of "hot spots." The last requirement arises because the sublimation rate of carbon- carbon increases by approximately a factor of 10 for each 100 K increase in surface temperature. Therefore, even localized areas with increased temperature can significantly contribute to the total mass loss rate of the heat shield. This consideration (and the desire to minimize distortion) also dictates that the shield material have a high thermal conductivity in order to minimize temperature gradients.

At first consideration, it would seem that a reflective surface (at least for the side of the thermal shield facing the sun at perihelion) would be an advantageous material selection in order to minimize the absorbed radiant energy. This approach has been rejected because of the unknowns associated with the response of reflective surfaces to the charged--particle, ultraviolet radiation, and micrometeriod fluxes that the spacecraft is exposed to during the long flight time. Unfortunately, determination of the material response to these environmental conditions is very difficult to simulate with ground tests that, of necessity, must be accelerated. However, surface modifications to reduce the shield temperature by increasing the hemispherical emissivity are being considered to provide an extra margin of safety in the thermal design.

Shield Technology Development Plan

The heat shield subsystem technology challenges will be addressed in a three phase program: (1) technology development related to the thermal and mechanical performance of the thermal shields including the fabrication development of carbon carbon materials , (2) technology development related to the RF performance of the primary thermal shield functioning as the spacecraft high gain antenna, and (3) shield system design, fabrication demonstration, and qualification. An integrated plan for these three efforts that supports launch of the spacecraft in 2001 has been developed. The specific technology development tasks include experimental testing to determine the mechanical, electrical, and optical properties, the mass loss rate, and the Rl reflectivity of carbon-carbon materials as a function of materials and process variations and temperature. In addition, special surface treatments designed to reduce the operating temperature and/or the mass loss rate will be investigated. The effect of the interplanetary radiation environment on the surface optical properties of selected carbon- carbon materials will also be investigated. Using the material characterization data generated in the initial phases of development, a preliminary materials selection will be made and the materials and process technology required to fabricate the primary heat shield and the high temperature secondary (infrared, IR) shields will be developed. Preliminary structural analysis to delineate the range of heat shield thicknesses required to survive launch loads will be included. In the final effort the manufacturability of the full-scale thermal protection system elements will be demonstrated. The following technology areas have been identified for investigation.

The optical properties of carbon-carbons including their dependency on material type (fibers, processing, and surface finish), temperature, wavelength, and angle of incidence will be characterized (Ayon, 1995). Material samples from JPL, NASA Langley Research Center, SAIC, BFGoodrich, Fiber Materials, inc., and Carbon-Carbon Advanced Technologies, inc. will be tested to determine optical properties with the emphasis on directional solar absorptivity and spectral hemispherical emissivity at temperatures up to 2400 K. The carbon- carbon test specimens will include samples with variations in fiber, weave geometry, processing, and surface treatment and/or coating. For instance, one material that will be characterized has been processed with a chemical vapor deposition (CVD) pyrolytic graphite coating for the possible reduction of solar absorptivity and mass loss rate. Other planned variations include the selection of high thermal conductivity fibers such as the Amoco K1 100 pitch fiber.

Special optical surfaces for potential carbon-carbon materials used for the heat shield will also be investigated. One example of surface modification will be provided by NASA Lewis Research Center using exposure to atomic oxygen to modify the micro-roughness of the surface. Because the scale of the surface modifications may be of the same order as the depth of material lost during perihelion passage, optical measurements on the surfaces are planned for both before and after exposure to a simulated perihelion passage. The objective of this work is to determine if the potential exists for surface modifications to sufficiently lower the heat shield maximum temperature to warrant the added complication and cost of using these methods on the spacecraft thermal shield.

The planned optical properties characterization will be carried out in US facilities in order to support a low cost and timely test program. These data will supplement previous data obtained in the CNRS Solar Furnace facility in Odeillo, France, using direct, concentrated solar insolation to heat the test samples (Robert, et al, 1995) . Other limited data has been obtained in US facilities using ohmic resistance heating or radiation from a halogen arc lamp (Randolph, 1991). Although these previous data clearly demonstrate the feasibility of the carbon- carbon thermal shield concept, they are not adequate to determine the dependency of key design properties on material parameters and to reduce design uncertainties to acceptable levels. The final objective of this work is to characterize the optical properties of thin carbon-carbon materials for use in design and to identify materials/surface finishes that may produce lower heat shield temperatures and minimum mass loss.

Key thermomechanical properties including in-plane tensile and compressive strength and elastic modulus, inplane shear strength, interlaminar shear strength and tensile strength, thermal expansion, thermal conductivity, specific heat, and bulk density will be characterized for the same material set as that used for tile optical properties characterization. The initial mechanical properties testing will be carried out by NASA Langley Research Center (Vaughn, 1995) at temperatures up to 1900 K. The thermophysical tests will be performed at US facilities and will include measurements to 2400 K. These data will supplement the optical characterization data in the selection of the final heat shield materials. After material selection is completed a more extensive thermomechanical material characterization will be performed to determine the properties required to complete final design of the thermal protection subsystem and to develop the optimum light weight design.

Determination of the mass loss rate of carbon-carbon as a function of temperature, material properties, and surface finish presents a technology challenge due to the extremely low rates of vaporization that occur for carbon materials below 2400 K. The vaporization rate of carbon is known to be a function of the' degree of graphitic versus amorphous structure and the orientation of the crystalline structure at the exposed surface (Randolph, 1991). The objective of this work is to verify the vaporization rate of carbon-carbon used for design and to identify potential materials/surface finishes that may produce lower heat shield mass loss rates than currently assumed. Although the vapor pressures of the primary gas phase species of carbon (Cl through C5) have been fairly well established, their vaporization coefficients as a function of temperature arc less well known. The current accepted values for the vaporization coefficients arc assumed to be independent of temperature and result in a reduction in mass loss rate by about a factor of 10 compared to that predicted using unity vaporization coefficients. Thus, it is important to confirm the vaporization rate to reduce risk.

The planned use of the carbon-carbon primary thermal shield as the high gain antenna for the spacecraft introduces a significant technical challenge that has not previously been investigated. Although carbon-carbon and other graphitic materials such as bulk graphite have been used, commercially for resistance heating elements in, high temperature furnace, little is known about the variation of its electrical properties as a function of temperature and their dependency on material constituents and manufacturing processes. Carbon-carbon has also been used for spacecraft antennas, but its RF reflective properties in the X-Band have not been systematically explored at high temperatures. Another issue concerns the thermionic emission from the carbon-carbon at the high shield operating temperature during perihelion passage. The thermionic emission chat acteristics of carbon-carbon are believed to be similar to those of tungsten. Tests are planned (Ayon, 1995) to determine these properties under realistic high-temperature conditions similar to those that will be encountered by the spacecraft.

The design of the shield system for the solar probe spacecraft relies on the use of low thermal conductivity secondary shields to block infrared reradiation from the primary shield from reaching the spacecraft bus. These secondary shields are planned to be fabricated from low density carbon and graphite felt or foam because of the high temperature reached by the front face of the secondary shield exposed to direct reradiation from the primary shield. This represents a technology challenge to develop, characterize, and demonstrate these secondary shield materials. The materials and processes required for manufacture will be investigated and developmental materials will be characterized to verify their predicted thermal conductivity. in addition, selected mechanical properties such as flexure modulus, flexure strength, and c oppression strength will be determined.

The final shield technology task will develop and demonstrate the manufacturing processes for the full-scale carbon-carbon materials required for the thermal protection system including the primary and IR shields and the associated structural details such as the support tubes joints and fasteners. It is anticipated that the required facility and test techniques will have been previously developed during the generation of the initial characterization data. The objective of this task is to sufficiently develop, characterize, and demonstrate the materials for the thermal protection system such that the flight shield design and subsequent manufacturing may proceed with confidence.

POWER TECHNOLOGIES

The spacecraft design constraints of using no radioisotopes for power generation or thermal control combined with the limited mass and volume available results in the incorporation of a number of advanced technologies into the power subsystem . The power subsystem concept includes a large, low mass solar array to deliver power from launch outward to Jupiter at 5.2 AU and then inward to approximately 0.7 AU , a smaller high temperature, high solar intensity array for power between approximately 0.7 and 0.3 AU, a secondary battery for off-sun and emergency power, a primary battery for power between approximately 0.3 AU and perihelion and the power management and distribution system to control all of these changing energy sources and spacecraft loads.

The area of the large solar array will be determined by the performance of the cells at 5.2 AU where 60 W electric are currently required for the spacecraft to survive. Solar cells have been tested in the laboratory under the Low Intensity Low Temperature conditions expected at Jupiter but solar arrays have not been demonstrated in space beyond the orbit of Mars (1.5 AU). Currently available single junction cells are assumed for this application instead of the more technologically advanced (and higher risk) multifunction solar cells. Either Si or GaAs/Ge cells will be candidates and the final selection will be based on overall mass, cost and performance. The technology development issues that remain relate to the lifetime and performance of the array over the continuous variation in temperature, solar intensity and angle of incidence of the sun on the cells as a result of feathering the array to assure survival. A significant testing program is underway to determine the best technology of solar cells to satisfy the extreme requirements of the mission. The solar array substrate is assumed to be a flexible fold out design based on the Advanced Photovoltaic Array developed by TRW (Kurland, 1994) under contract to JPL. The structure and deployment of the array assumes a low mass inflatable rigidizable technology that was ground tested in vacuum by L'Garde (Davey, et al, 1994). A fall back to this design would be a more conventional hi-stem. Independent of the structural and deployment approach selected, the array must be articulated to properly point the array off the solar normal direction to compensate for the large variation in solar intensity on this mission. This array would be jettisoned at about 0.7 AU or when its temperature reaches about 400K.

The smaller high temperature solar array will supply power from about 0.7 to 0.3AU and will use a rigid substrate with GaAs/Ge technology cells welded together to maximize performance and survivability. The deployment and pointing functions required by this array will probably be performed with the same mechanism as the larger array.

Commercial y sponsored testing is currently being conducted at JPL on a number of cell designs to determine the effects of temperature, angle of solar incidence and solar intensity on cell performance. These results should be extremely valuable in determining the optimum cell design.

A secondary battery provides power during launch prior to deployment of the solar array, during off-sun point activities such as during trajectory correction maneuvers and for meeting spacecraft peak power demands. The high energy density and low self discharge rate Li-ion battery candidate for this mission is superior to the more conventional Ni chemistry batteries currently available. The lower cycle life of the Li-ion battery compared to other Ni chemistry batteries should not be a major concern due to the relatively low number of cycles expected on this mission. One area of uncertainty for the Li-ion battery is in the active shelf life which is currently greater than 5 years but still may be a concern for this mission.

Primary batteries using Li-SOCl2 cells were selected over the more conventional Li-SO2 cells due to the approximately 50% higher energy density. The two are comparable in all other areas like operating temperatures, shelf life and self discharge rate with the exception of the relative experience both for terrestrial and space applications where the Li-SO2 has a clear advantage. Fuel cells were considered and rejected for this mission because of mass and other technology development concerns.

CONCLUSIONS

The technology challenges of the Solar Probe mission are significant! but expected to be resolved at the completion of the technology development program that is now underway. Major tasks are to be completed in 1995 and 1996 which will enable these technologies to be incorporated into a design concept leading to a flight prototype. Shield materials tests will be completed in 1996 to allow, for example, a selection of a specific carbon-carbon material for the primary thermal shield. Solar cell tests will also be completed that will allow a final design concept for the solar array(s) to be determined. In addition, in 1996 the development of miniaturized scientific instruments will begin with the release of a NASA Research Announcement (Howard, Randolph, 1995) for Solar Probe instrument technology development.

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